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Internal Note No. 67-FM-175

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC INTERNAL NOTE NO. 67-FM-175

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PROJECT APOLLO

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Approved: ______________________
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SUMMARY

The mission trajectory design for the NASA manned space-flight programs has been tailored with flight safety as a primary consideration. Various types of constraints, which are basically classified as hardware, software, and operational, have been considered in the mission design for these programs. The impact on mission design of these constraints is discussed, and specific examples that have affected the Mercury, Gemini, and Apollo programs are used as a basis for the discussion. In addition to the impact on the mission trajectory design, these types of constraints also require considerable mission analysis to define flight control limits, mission rules, and operational procedures. These limits and procedures are used in real time by the flight crew and the flight control team to further enhance flight safety. Specific examples are discussed in this regard for the various mission phases. Similarity and carryover from one program to another are pointed out. It is shown that, through the application of proper mission design, constraint definition, trajectory control limits, and operational procedures, flight safety for manned earth orbital missions has been achieved.

INTRODUCTION

During all of the United States civilian manned space-flight programs, flight safety has always been the prime factor. Mission design has, therefore, always had as a goal the maximizing of this factor. Flight safety and crew safety are synonymous as far as mission design is concerned. A satisfactory mission design is thus one that accomplishes the required mission objectives within the known constraints and minimizes the risk to the crew. The problem of how this has been accomplished, what contraints have been considered, and how these constraints have affected mission design is the subject of this paper.

MISSION DESIGN

A definition of what is meant by mission design is depicted in figure 1. Basically mission design consists of two prime functions. The first function is the analysis and coordination necessary to arrive at an acceptable nominal mission design and the associated alternate mission plans. The second prime function is in the area of contingency analysis, which is mainly concerned with flight control, operational procedures, and real-time "what-if" problems. In both cases, mission objectives and flight safety must be satisfied within the known constraints. The constraints are basically associated with hardware (launch vehicle, spacecraft, and ground), operational procedures (flight control, recovery, and crew), and software (onboard and ground). The nominal mission design is needed to supply the planning activities shown on figure 1, and, of course, bears on the contingency analysis. The contingency analysis is all the work necessary to answer the questions in the operations areas shown on figure 1. Thus, the term mission design, as used at the Manned Spacecraft Center, refers to all the effort necessary to satisfy both the planning and operational areas for each mission.

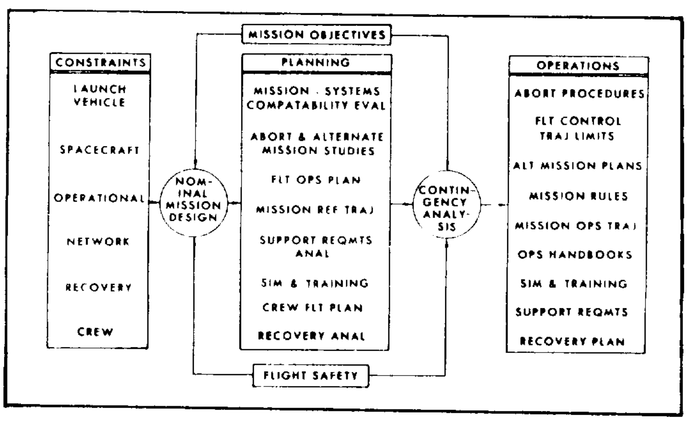

Figure 1.- Mission design procedure.

The constraints that affect nominal mission design, as would be expected, are somewhat different from those that affect the contingency mission planning. The prime constraints that affect each area of mission design are listed in table I. The nominal mission design is primarily constrained by orbital altitude, usually perigee and, to a lesser extent, apogee; orbital inclination; and descent techniques. The contingency mission planning is primarily concerned with abort procedures and flight control trajectory limits. These constraints and considerations, in both areas of mission design, are usually classified as operational, configuration and systems (hardware), and software.

TABLE I.- MISSION DESIGN FOR FLIGHT SAFETY

Nominal Mission Design
Orbital Altitude
Orbital Inclination
Descent Techniques
Contingency Mission Planning
Abort Procedures
Flight Control Trajectory Limits
Constraints and Considerations
Operational
Configuration and Systems
Software

The various aspects of these constraints and considerations that have affected mission design and flight safety are listed in table II. The operational constraints are generally associated with flight control and recovery capability (voice communications, command capability, and monitoring) crew and ground personnel procedures and training, and environmental aspects. The configuration and systems constraints enter into the mission design in almost every area. This is especially true of the spacecraft. The launch vehicle, generally speaking, does not affect the nominal mission design from a safety point of view, but does enter into the contingency analysis as will be pointed out later. As can be seen from the spacecraft constraints listed in table II, the nominal, as well as contingency, mission design will be affected by one or more of these constraints. The software constraints have not been critical from a safety and mission design standpoint. They are certainly a consideration and, depending on the specific capability, can give a greater degree of confidence or assurance of crew safety. Through proper design, software can increase the ground control and aid to the crew, and permit onboard capability for certain navigation and maneuver computations. This obviously increases crew safety but does not significantly affect mission design.

The effect of these various constraints and considerations on the mission design for the three NASA manned space-flight programs can best be discussed by considering the nominal mission design and contingency mission design phases separately.

TABLE II.- CONSTRAINTS AND CONSIDERATIONS

Operational
<u>Landing and Recovery</u>: Geographic, Lighting, Communications, Logistics, Medical Support
<u>Communications and Tracking</u>: Systems Monitoring, Ground Command Capability

TABLE II.- CONSTRAINTS AND CONSIDERATIONS - Concluded

Environmental Surroundings: Atmospheric Properties, Winds, Lightint, Weather Conditions, Radiation, Meteors
Human Factors: Crew Acceleration and Deceleration Tolerances, Crew and Ground Response Times
Procedures: Separation Techniques, Recontact Avoidance, Simple and Reliable for Training Proficiency, Mission to Mission Carryover
Range Safety: Land Impact Avoidance, Orbital Collision
Configuration and Systems
Launch Vehicle
Propulsion: Emergency Detection, Switchover
Guidance and Control: Emergency Detection, Switchover, Stability
Structural: Emergency Detection
Spacecraft
Propulsion: Type, Performance, Backup Capabilities
Guidance and Control: Performance, Procedures, Backup Systems
Structural: Couch Supports, Landing
Thermal (Heat Shield): Heating Limitations, Space Soaking
Aerodynamics: Stability, Trim and Lift/Drag Characteristics
Window-Crew Geometry: Crew Visibility to Horizon, Manual Takeover
Consumables: Electrical Power, Environmental System, Propulsion
Sequencing: Attitude Requirements, Time, Procedures
Software
Launch Vehicle: Ground Control Capability, Up/Down Data Capability
Spacecraft: Up/Down Data Capability, Systems Evaluation
Ground Control: Up/Down Data Capability, Systems Evaluation

NOMINAL MISSION DESIGN

Of the three major factors stated earlier (table 1) that affect nominal mission design, orbital altitude and orbital inclination have been the most constraining. The third factor, descent techniques, basically is coupled to the acceptable minimum perigee altitudes which come under the orbital altitude considerations. The orbital altitude constraint is made up of the considerations as shown in figure 2. In the high altitude region (above about 400 n. mi.), radiation is a prime consideration.

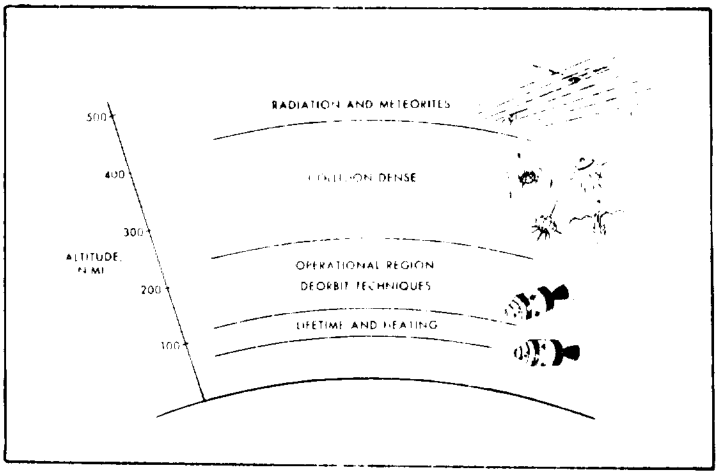

Figure 2.- Orbital altitude considerations.

Meteorites, although a consideration, are not nearly as much of a mission design constraint as is radiation. Neither consideration sets an absolute limit on the apogee altitude but does establish a region that is avoided if possible. In the intermediate altitude region between approximately 225 and 400 n. mi., the consideration is collision with other spacecraft or parts thereof. Again this consideration does not establish an absolute limit but a region to be avoided if possible.

Perigee altitude is limited by mission lifetime requirements and spacecraft heating caused by passage through the atmosphere. These considerations establish a limit above which the mission orbital altitudes must be designed. The region between the undesirable lower and two upper regions is the operational region. If at all possible, the mission orbital altitudes will be designed to stay within this region which extends from approximately 100 to about 225 n. mi.

Descent Techniques

The lower limit of this operational region is established by the capabilities of the descent techniques. Descent technique means the type of propulsion system, available backup system, sequencing, aerodynamic capability, procedures, and so forth. The configuration of the spacecraft systems used for descent for each manned spacecraft program has been different, as seen in figure 3. The Mercury spacecraft was the simplest in that all systems except for the retrorockets were carried inside the entry module. In the case of Gemini and Apollo spacecraft, only the systems necessary for entry are carried in the entry module. In the Gemini spacecraft, the orbital equipment adapter was jettisoned prior to retrofire, and the retrorocket adapter was jettisoned after retrofire. In the Apollo program, the service module (SM), which contains the propellant, main engine, reaction control system (RCS), and other orbital systems, is jettisoned after the deorbit maneuvers. A comparison of some of the characteristics of the descent systems for each program is shown in table III. The Apollo spacecraft is the first to use liquid propellant and has an unlimited deorbit capability compared to the Mercury and Gemini spacecraft. Any failure of the single engine Apollo system will result in the use of a backup propulsion system for deorbit. In the case of the Mercury and Gemini spacecraft, the orbit altitude could be controlled so that one retrorocket misfire would still result in an acceptable entry.

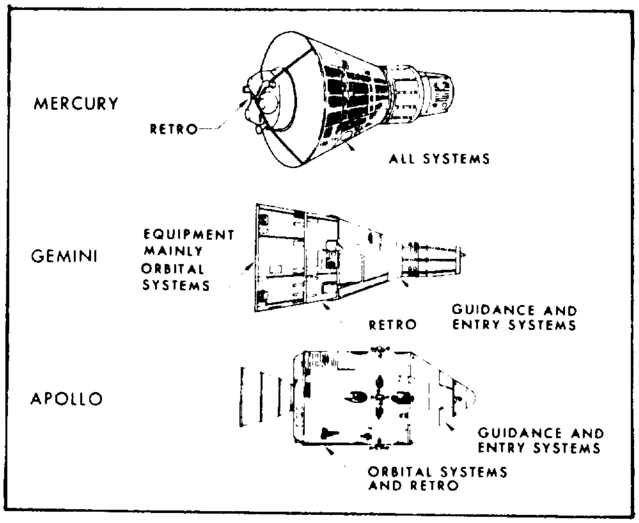

Figure 3.- Spacecraft configurations.

	MERCURY	GEMINI	APOLLO
PRIMARY DESCENT IMPULSE SYSTEM			
PROPELLANT TYPE	SOLID	SOLID	LIQUID
NO. OF ENGINES	THREE	FOUR	ONE
DEORBIT ALTITUDE CAPABILITY			
ALL ENGINES FIRE	≈200 N MI	≈200 N MI	——
ONE ENGINE MISFIRE	≈100 N MI	≈100 N MI	BACKUP SYSTEM
BACKUP DESCENT IMPULSE SYSTEM			
PROPULSION SYSTEM	RCS	OAMS AND RCS	SM AND CM RCS
DEORBIT ALTITUDE CAPABILITY	≈100 N MI	≈100 N MI	≈100 N MI
AERODYNAMICS			
LIFT DRAG RATIO	ZERO	≈0.18	≈0.28
SEQUENCING			
JETTISON TECHNIQUE	SPRING	SPRING	SM RCS
RECONTACT EFFECT	NEGLIGIBLE	SLIGHT	SLIGHT
THERMAL PROTECTION	ADEQUATE	ADEQUATE	EXCELLENT
CONSUMABLES			
POST MANEUVER	LIMITED	LIMITED	AVAILABLE
ENTRY INTERFACE	LIMITED	LIMITED	LIMITED

TABLE III.- DESCENT SYSTEM CHARACTERISTICS

The backup systems available for each program for the deorbit maneuver are the entry reaction control systems. In addition, the Gemini orbital attitude maneuvering system (OAMS) and the Apollo service module reaction control system (SM RCS) are available and capable of performing a deorbit maneuver. All of these backup systems are most efficiently used at apogee, and thus result in reduced control over the choice of the landing point in comparison to the primary propulsion system which permits unlimited choice of the landing point. In all cases the backup entry RCS are capable of deorbit from approximately 100 n. mi. The Gemini OAMS and the Apollo SM RCS increased this capability considerably. One disadvantage of the Gemini OAMS, which is not a problem with Apollo, is that the OAMS had to be used prior to using the primary retrorocket system. This was because the OAMS was jettisoned with the equipment adapter, as was mentioned earlier, prior to firing the retrorockets. Thus, the basic advantage of the Gemini spacecraft over the Mercury spacecraft was the maneuvering ability to maintain the orbital altitudes within the desired limit, considering the three-out-of-four entry capability and the confidence in the retrorocket system. The Apollo spacecraft has this same maneuvering advantage in addition to the capability to use the backup system after attempting use of the primary system. The Gemini and Apollo spacecraft have a limited aerodynamic lift modulation capability which increases their capability to insure capture and entry.

Other descent system characteristics which influence mission design and constrain the orbital altitudes are the sequencing of events prior to and following the deorbit maneuver, the thermal protection system, and the available consumables. The sequencing has generally been concerned with the avoidance of recontact with the jettisoned items. The jettison technique used for the Apollo program offers the possibility of achieving the greatest separation distances since it uses the SM RCS propulsion capability for separating the service module from the command module (CM) or entry module. However, because of the size, shape, and aerodynamics of the SM, and the sequence of events that must be followed in order to conserve the consumables, recontact is more of a problem on Apollo missions than it was on Gemini or Mercury missions. On Gemini missions, the sequencing was of concern mainly because of the size of the jettisoned equipment adapter and the fact that consumables were contained in this item. The equipment adapter remained in orbit which minimized recontact problems. In all the programs, the sequencing had to be such as to allow enough time for proper orientation of the spacecraft for entry and for use of alternate systems for jettisoning the necessary items.

In the Mercury and Gemini programs, the thermal protection system has been certainly adequate and, in the case of Apollo earth orbital missions, is excellent since the system is designed to operate at lunar return entry conditions. The consumables in all programs have been, and are, more than adequate but limited since, except for Mercury missions, a large part of the consumables were jettisoned prior to entry. The consumables of concern are the environmental and electrical power consumables. Proper design of the entry techniques and development of proper crew procedures have avoided any problems, assuming nominal system performance. Systems degradation or failure obviously can present a problem, but redundancy, backup systems, and crew procedures have reduced this problem to an acceptable level.

Orbital Inclination Considerations

The final major constraint that affects nominal mission design is the orbital inclination. Once the spacecraft has achieved orbit, it must be ascertained that the systems are performing acceptably and that the orbit is acceptable to accomplish mission objectives. This usually requires several revolutions around the earth and proper voice, telemetry, command, and tracking coverage from the ground. If the first few hours in orbit reveal any reason for early termination of the mission, it is also desirable to have the orbital ground traces pass over or near established recovery areas. In addition, generally speaking, orbital inclinations greater than about 35° or 40° result in emergency landings in frigid water, which may be hazardous to crew survival.

Thus, for all three manned space-flight programs, the orbital inclinations have been, and are, restricted to values less than 40° and the flight azimuths at launch have been, and are, between approximately 70° and 90°. The reasons for these restrictions are depicted in figure 4. The ground traces for the first five or six revolutions for three flight azimuths at launch are shown on a mercator projection of the earth along with the current ground stations used for earth orbital missions. For the 70° flight azimuths, the ground coverage is very good for the first four revolutions. As the azimuth increases from 70° to 110°, the ground coverage decreases until at 110° azimuth the coverage is good only for approximately one revolution. In addition, the ground traces pass over or near the prime Atlantic recovery area for the first three revolutions at the 70° flight azimuth. At the 110° azimuth, none of the early revolutions pass near the Atlantic recovery area. This is important since the launch usually occurs in daylight and the prime Pacific area is therefore in darkness. Thus the northerly flight azimuths permit daylight recovery in the Atlantic and generally reduce the possibility of or eliminate a landing at night in the prime Pacific

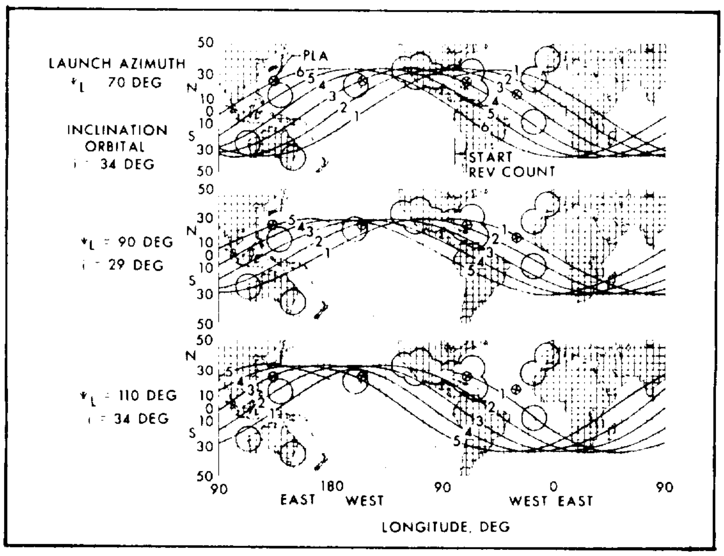

Figure 4.- Orbital inclination considerations.

recovery area, since the prime Pacific recovery area is approaching daylight caused by the rotation of the earth. Thus the lighting situation improves the longer the Atlantic area is available.

Other considerations which affect flight safety and mission design are the mission requirements for large propulsive maneuvers (on the order of several thousands of feet per second), consumables management, and lighting during the orbital phase. Large propulsive maneuvers must be directed out-of-plane to remain within the orbital altitude restrictions. They, also, must usually be placed over ground stations to provide ground monitor and assistance. This type of maneuver has the advantage of changing the line of nodes of the orbit such that the ground traces move eastward, thus increasing ground coverage and landing capability. The consumables management consideration is one of assuring sufficient propellant and other systems lifetime for a safe entry and landing. The orbital lighting consideration affects the timing and location of rendezvous and docking and thus can restrict the orbital period or orbital altitude.

CONTINGENCY MISSION PLANNING

During the mission design phase, as has been discussed, the flight profile begins to evolve from the series of iterative analysis cycles made by the mission designer among the mission constraints, flight objectives and requirements, configuration and systems capabilities, and operational support factors. As the flight profile evolves and mission details begin to fall into operational sequences, a more rigorous and sophisticated analysis starts in the area of contingency mission analysis. The prime consideration in this area of contingency mission design is to assure flight safety by developing abort procedures, flight trajectory limit lines, mission rules, and final flight plans, and by establishing operational procedures. This is done in conjunction with the mission operations team which consists of the flight crew, ground controllers, and recovery forces. Experience thus far in manned space-flight programs has shown, although it is probably not generally known, that the largest part of the mission design effort is spent in the design and analysis of contingency mission planning.

The establishment of abort procedures and trajectory control limit lines are the two areas of contingency mission planning which predominately affect flight safety. This effort is usually related almost exclusively to those mission phases which can be defined as being "time critical." "Time critical" is used to imply the need for a fast action response from the flight crew and ground control team to in-flight failures or malfunctions which would cause the trajectory to deviate into a flight regime which would constitute a danger to the crew and sometimes earth inhabitants. To plan and be prepared for these situations, the mission designer must investigate numerous off-nominal performance characteristics, and, in addition, the many "what-if" type of questions which arise during the planning of a mission. It has been established that the most reliable procedure to use during an emergency is usually that procedure which is the simplest and easiest to perform. This does not necessarily make the optimum use of all systems and operational capabilities. Nevertheless, this philosophy has been followed successfully in all of the NASA manned space-flight programs.

For convenience, the mission will be broken down into three basic phases - ascent, orbital, and descent. The effects of the constraints listed in table II on each phase will be discussed and examples will be given for each program.

Ascent Phase

The ascent phase of a typical mission is shown descriptively in figure 5. This figure illustrates the various flight regimes through which the space vehicle must fly to obtain orbital flight. These flight regimes are atmospheric, transition, and space, and are a major consideration in the design of the ascent phase abort modes. The atmospheric region is considered to extend from the earth to a 280 000-ft altitude which has been the approximate altitude at which reentering spacecraft have begun to sense appreciable deceleration forces. The space regime is considered to be above an altitude of 400 000 ft, with the region laying between the atmospheric and space regions being the transition region between atmospheric and space-flight conditions, When the constraints and considerations as detailed in table II are considered in combination with the three flight regimes and spacecraft performance capabilities, abort modes and associated trajectory control limits can be defined.

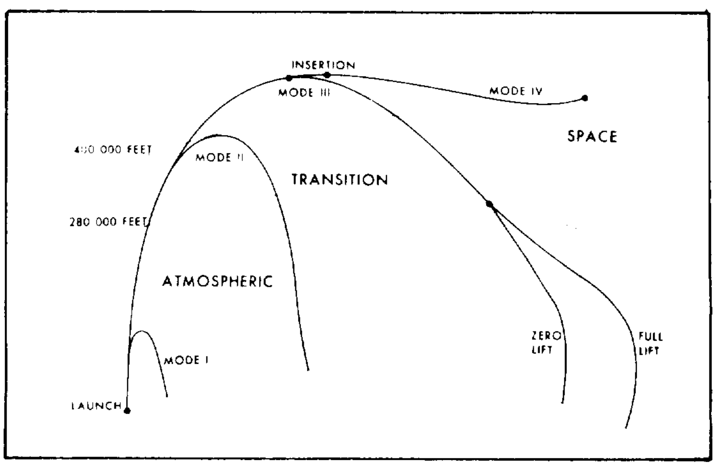

Figure 5.- Flight regimes and abort mode definition.

The first abort mode involves an abort which is almost always in the atmosphere and lands a relatively short distance from the launch area. This mode has to have specially developed systems for separation from the launch vehicle because of the time critical factors of impending explosions of the launch vehicle and the aerodynamic forces acting against the escaping vehicle.

The second abort mode region is a transition type of abort which is initiated essentially out of the atmosphere but has a descent back into the atmospheric region. This mode usually covers the major portion of the ascent phase and is the most simple and reliable abort procedure. The launch vehicle explosion and stability factors have lessened, thus allowing a less critical separation technique. For past programs, this separation technique has been the nominal orbital separation procedure. In addition, since the entry velocities are not too great the spacecraft has no footprint control capability and the entry procedure is designed only for deceleration force alleviation.

The third mode of abort is one of the most critical and involves the use of the spacecraft propulsion and aerodynamic capabilities for landing area control since the ground track of the instantaneous landing point begins to move rapidly toward an orbital track around the earth.

The fourth mode of abort is defined as the contingency orbit insertion mode. As insertion conditions are approached, the spacecraft propulsion capabilities can be used to obtain an acceptable orbital insertion. This mode is a primary mode of abort, since once orbit is obtained, the flight crew and ground controllers can assess the situation and either perform some alternate mission plan or prepare to deorbit at the end of the first orbit into a prime recovery area.

The major operational and configuration and systems factors which are considered in each of the abort modes and those that also have a primary affect on the design of the trajectory control limits for the ascent phase are shown in table IV. The primary operational factors for the Mode I aborts are the environmental surroundings and procedures. The main constraint is the spacecraft propulsion and sequencing, since the spacecraft must separate and escape from the area of a thrusting launch vehicle which could explode. In addition, the spacecraft is flying in a region of high aerodynamic forces and both land and water landings are involved. Mode I utilized, and utilizes, the escape tower configuration for Mercury missions and for Apollo missions; for Gemini missions, ejection seats were used at low altitudes and a ride-out technique at the higher altitudes.

The Mode II abort is the simplest and most reliable, and covers the longest period of time. This mode is basically constrained by the human factor of the crew being able to withstand the entry deceleration

	Ascent Abort Mode Procedure I	II	III	IV	Flight Control Trajectory Limits
● OPERATIONAL					
• LANDING AND RECOVERY			X		X
• COMMUNICATIONS AND TRACKING			X	X	
• ENVIRONMENTAL SURROUNDINGS	X				
• HUMAN FACTORS		X		X	X
• PROCEDURES	X		X	X	X
• RANGE SAFETY					X
● CONFIGURATION AND SYSTEMS					
● LAUNCH VEHICLE					
• PROPULSION					X
• GUIDANCE AND CONTROL					X
• STRUCTURAL					X
● SPACECRAFT					
• PROPULSION	X		X	X	X
• GUIDANCE AND CONTROL		X	X	X	X
• STRUCTURAL			X		
• THERMAL (HEAT SHIELD)			X		X
• AERODYNAMICS		X	X		X
• WINDOW CREW GEOMETRY			X	X	
• CONSUMABLES			X		
• SEQUENCING	X	X	X	X	X

TABLE IV.- ASCENT PHASE CONSIDERATIONS

loads. Therefore, the spacecraft aerodynamic and guidance and control characteristics are used to alleviate the crew entry loads. Spacecraft sequencing is utilized for separation from the launch vehicle. As stated previously, the Mode II abort consists of very simple but reliable procedures which require only that the spacecraft obtain adequate separation, jettison unnecessary equipment, orient to the entry attitude, and fly a simple entry procedure. A rolling entry was used in Mercury, and the Gemini and Apollo modes used, and use, full lift to reduce the entry loads on the crew. In addition, there is no landing area control for this mode.

The Mode III abort is utilized during the latter part of the ascent phase to control the landing area and is required primarily to avoid land landings in Western Africa. The primary operational factors for this mode are landing and recovery, communications and tracking, human factors, and procedures. Communications and tracking is a constraint because the ground has to pass data to the spacecraft for the required propulsion maneuver and time delay until firing for landing area control. Human factors and procedures are a constraint because of time delays in processing tracking data, computing, and then passing the data to the spacecraft. In addition, certain procedures are required to avoid recontact with jettisoned components of the spacecraft. All of the spacecraft configuration and systems factors affect this mode. The window-crew geometry constraint is of primary importance as it is used to check or backup the spacecraft attitude reference orientation prior to maneuvers and at entry. The procedures capability, the sequencing, the propulsion maneuver, and consumables affect the type of descent profile that can be flown. The guidance and control and aerodynamic factors are utilized to correct landing area control accuracy. The thermal and structural capability also is a consideration as entry conditions become significant. The mode is the most complicated and difficult to design because of the time critical factors.

The Mode IV abort, or the contingency orbit insertion, is operationally constrained by communications and tracking, human factors, and procedures. All these constraints are because of the requirement for a very accurate propulsion maneuver needed to obtain acceptable orbital conditions. Again, as in the Mode III aborts, communications and tracking information must be passed from the network to the Mission Control Center where the data are analyzed; maneuver and time sequence information must then be passed to the crew. The primary spacecraft factors are the propulsion capabilities, sequencing, guidance and control, and window-crew geometry. The sequencing is a constraint because of the timing and operations required by the crew to perform the spacecraft maneuver, to reorient to the required insertion maneuver attitude, to perform the insertion maneuver, and then to evaluate the postmaneuver orbital conditions. Window-crew geometry is a constraint because the insertion maneuver attitude is constrained to those attitudes at which the crew can use visual sightings of the horizon of the earth for their check on attitude reference and as a backup to possible systems failures. This area of mission design has required many extensive simulations and training exercises with the crew and ground control personnel to validate the Mode IV design procedures.

To assure crew safety during the ascent phase, it has been necessary to design automatic abort systems as well as to establish trajectory control limit lines and abort procedures. The automatic abort systems in each of the programs thus far have been designed to provide for crew safety for those failures or malfunctions which require time critical aborts. The trajectory control limit lines and abort procedures are designed to protect the crew from those failures or malfunctions which result in a slow drift and divergence of the actual flight trajectory from the planned nominal trajectory. These types of anomalies are not easily detected by the spacecraft onboard systems and therefore trajectory control limits are used to terminate launch vehicle thrust, and to abort or to switch over control from one vehicle to another. The operational factors which affect the design of the trajectory control limit lines are landing and recovery, human factors, procedures, and range safety constraints. The configuration and systems factors for the launch vehicle are the propulsion, guidance and control, and structural capabilities of the vehicle. The spacecraft factors are propulsion, guidance and control, aerodynamics, sequencing, and thermal protection capabilities.

The basic trajectory control limit lines that have been developed for the Mercury, Gemini, and Apollo programs are shown in figure 6. These examples show the trajectory control limits as functions of the inertial flight-path angle and velocity during the ascent phase. These display parameters have been found to be the most useful in monitoring the ascent phase and are considered to be the standard display. However, other trajectory parameters such as time, altitude, range, predicted landing points, acceleration, and so forth, in conjunction with launch vehicle and spacecraft telemetry information, are all evaluated with the standard display in real time by the ground controllers. During

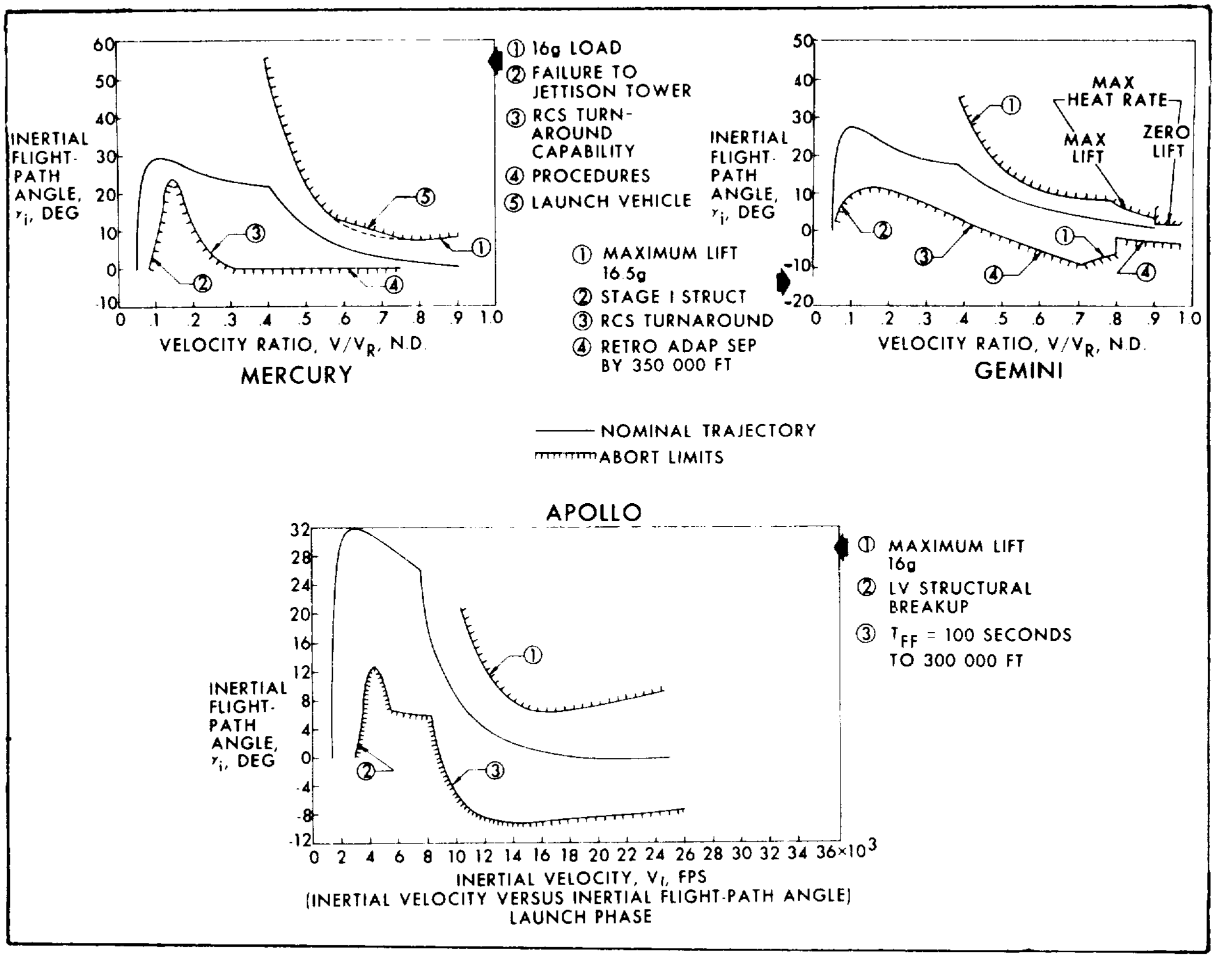

Figure 6.- Ascent phase trajectory control limits.

the Mercury program, a Mode I limitation was determined based upon the capability of the jettison rocket of the escape tower to separate the tower from the spacecraft. Since the apex portion of the spacecraft also housed the parachute system, it was necessary to insure that the tower jettison rocket always had the capability to overcome any aerodynamic forces it might encounter. The flight corridor was also constrained by a 16 g deceleration force limit for the crew during entry following an abort and by the capability of the spacecraft rate control system to counteract aerodynamic forces and orient the spacecraft in a heat-shield-forward attitude. Here again the intent was to protect the parachute compartment from aerodynamic loads and from heating which could result in parachute system failures. A procedures limit was established at the higher velocities to initiate an abort if the launch vehicle had diverged off course and had begun to descend.

For the Gemini program, the Mode I region was bounded by the launch vehicle structural breakup considerations and the Mode II region, as in the Mercury program, was bounded by a 16.5 g entry deceleration crew limit for the high flight-path angles and by the spacecraft rate control systems turnaround capability for the lower flight-path angles. In addition, a new limit was needed to bound the time of free fall required by the abort procedures in completing all of the required separation, jettisoning, and maneuver requirements dictated by the spacecraft configuration and systems. In the higher velocity region, the Mode III region was bounded by the thermal protection capabilities of the spacecraft.

For the Apollo program, the Mode I abort region is bounded by launch vehicle structural considerations, and the Mode II and III regions by the 16 g entry deceleration crew limit and the time-of-free-fall constraint to complete the abort sequencing.

The design of abort procedures and trajectory control limit lines for the Mode IV aborts or orbital insertion region are shown in figure 7. This figure is essentially a continuation of figure 6 with the scales blown-up to better illustrate the many considerations and procedures required during the final portion of the ascent phase. During the Mercury insertion, the spacecraft only had descent propulsion available which made the insertion monitoring simply the choice of whether an orbit had been achieved or not. If a no-go was determined, Mode III abort procedures for landing control would be passed the crew. Limit lines were determined to keep the launch vehicle from inserting the spacecraft in an overspeed or high apogee orbit because of spacecraft reentry heating and descent capability limitations.

For the Gemini program, the addition of the OAMS gave the spacecraft additional propulsion capability. Thus there was a region in which this system could be used to complete an acceptable orbit in case the launch vehicle had shut down early. This region was bounded at the higher flight-path angles by operational procedures, which defined a boundary based on efficient use of propellant either immediately or by waiting until apogee was reached to make the corrective spacecraft maneuvers. Again, the acceptable altitude considerations were based on heating and lifetime constraints. In addition, the display also shows the transition of abort capability to go from the Mode III to the Mode IV abort procedure.

The Apollo display is for the uprated Saturn IB ascent phase where additional spacecraft propulsion capability, equivalent to about 3000 fps is available. With this increase in spacecraft propulsion capability, the Mode IV region is appreciably increased. Again, as in the Gemini program, a boundary has been defined based on efficient use

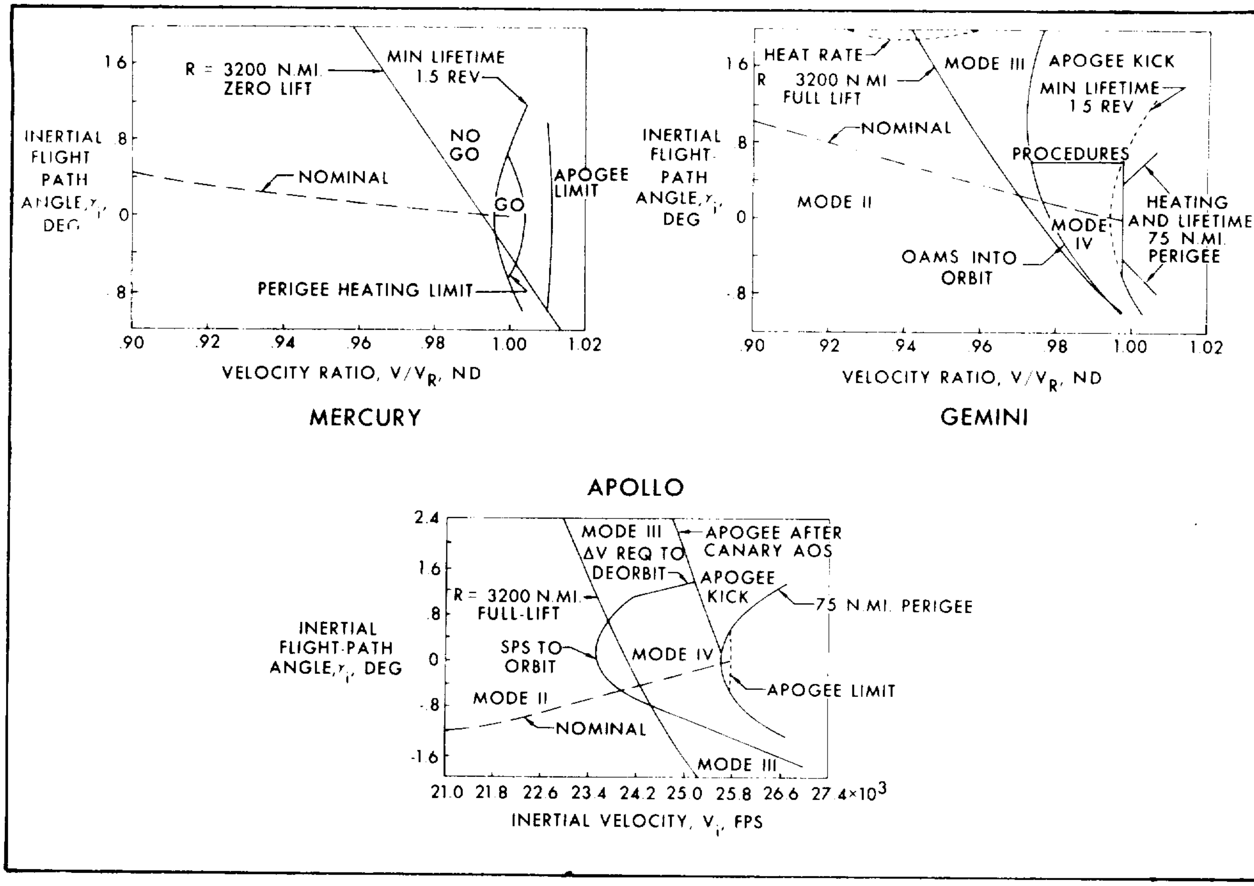

Figure 7.- Orbital insertion.

of propellant either immediately or at apogee. Ground communications considerations have also entered into this boundary. This display also shows the capabilities of the Mode III and Mode IV abort procedures. During both the Gemini and Apollo programs in which additional spacecraft propulsion capability was available, the limit lines have been biased to account for attitude and altitude dispersion errors and computational procedures.

Orbital Phase

The considerations that affect contingency mission design for this mission phase are listed in table V. The consideration is basically one of propellant and consumables management in order to reserve sufficient propellant for deorbit and attitude control during the deorbit maneuver and to maintain a perigee altitude consistent with backup deorbit capability. The criticality of systems failures must be considered in establishing descent procedures and maintaining proper trajectory control of ground traces to maximize emergency recovery capability. The lifetime, perigee, apogee, and communications considerations are generally taken into account during the nominal mission design. The specific analysis required to assure flight safety is concerned with determining absolute limits on the various trajectory and spacecraft parameters. Typical examples are the establishment of minimum perigee altitude (75 n. mi.), propellant and other consumables red lines, maneuver attitude limits to remain within altitude and ground track constraints, and emergency separation procedures when in a docked configuration.

TABLE V.- ORBITAL OPERATIONS PHASE CONSIDERATIONS

Altitude Operations Limited to Systems Capability
Maneuver Budget
Propellant Reserved for Deorbit
Backup Systems and Manual Takeover Capability
Criticality of Various Systems Failures
Lifetime Control
Perigee Altitude Heating Constraint
Apogee Altitude Environmental Constraints
Rescue Maneuver Capability
Communications and Tracking

The rescue maneuver capability consideration has become apparent in the Apollo program as a result of the requirement to perform manned test of the LM in earth orbit. The LM does not have any thermal protection for descent to the surface of the earth. It does, of course, have several propulsion systems (descent engine, ascent engine, and RCS). This rescue consideration on the Apollo earth orbital mission represents a severe constraint on the SM consumables as well as LM maneuvers and separation distances. A great deal of contingency mission design has gone into the planning of those Apollo missions which require manned operation of the LM undocked from the command service module and thus far it has not been necessary to sacrifice mission objectives to assure flight or crew safety.

Descent Phase Considerations

Experience has shown that this mission phase requires considerable effort for the design of the deorbit maneuvers, entry control, limits defining safe entry, and backup procedures. The constraints listed in table II which affect the various parts of this mission phase are shown in figure 8. The considerations which affect the deorbit maneuver, separation sequence, spacecraft orientation, and the free fall time from end of deorbit maneuver to the entry interface are given on the left side of the figure. For comparison, typical free fall times from deorbit maneuver to entry interface for each program are given in the upper right corner of the figure. These times cannot be less than some minimum value. For instance, the minimum free fall time required for Apollo missions is approximately 10 minutes. For Mercury and Gemini missions, the minimum time required was on the order of 3 to 5 minutes. These free-fall-time requirements are determined by the propulsion, attitudes, sequencing, and crew procedures during and following the deorbit maneuver. The deorbit maneuver itself is usually placed over or just before a ground station for tracking and communication reasons and is planned so as to have a lighted horizon for use as a backup out-the-window attitude reference. This has been the case for all three manned space-flight programs.

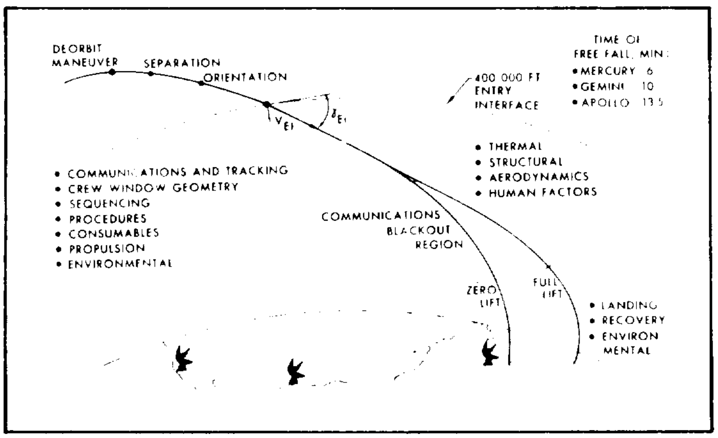

Figure 8.- Descent phase considerations.

The deorbit maneuver is designed to achieve a specific set of entry conditions (velocity and flight-path angle) at the entry interface (usually between 350 000 to 450 000 ft). These entry conditions are based on the considerations affecting the atmospheric part of the descent phase and the landing location. The atmospheric portion of the descent phase is constrained by those considerations shown near the center of figure 8. For earth orbital missions the most constraining factors have been the human factors (tolerance to deceleration conditions) and thermal considerations. The spacecraft structural and aerodynamic considerations must be considered in conjunction with the two prime factors. The aerodynamics capability can relieve or increase the deceleration levels experienced by crew and spacecraft and also can affect markedly the thermal environment (total heat and heat rate) experienced by the spacecraft. The entry must also be designed so that none of the spacecraft component structural constraints are exceeded (for example, crew couch supports, instruments, etc.).

A considerable portion of this phase is in the communications attentuation or blackout region. This is another reason for placing the deorbit maneuver over or prior to a ground station. The information gathered immediately after the deorbit maneuver and prior to communications blackout is most important to aid in predicting the probable landing point and relaying to the crew any backup entry information.

The selection of the landing location is constrained by geographic, logistic, communications, weather, and lighting considerations. It is also affected by all the other descent constraints in that these constraints determine the total range traveled from the deorbit maneuvers to landing.

To demonstrate how the constraints have affected the descent trajectory control limits, a comparison of the descent phase trajectory control limits for each manned program is shown in figure 9. The control

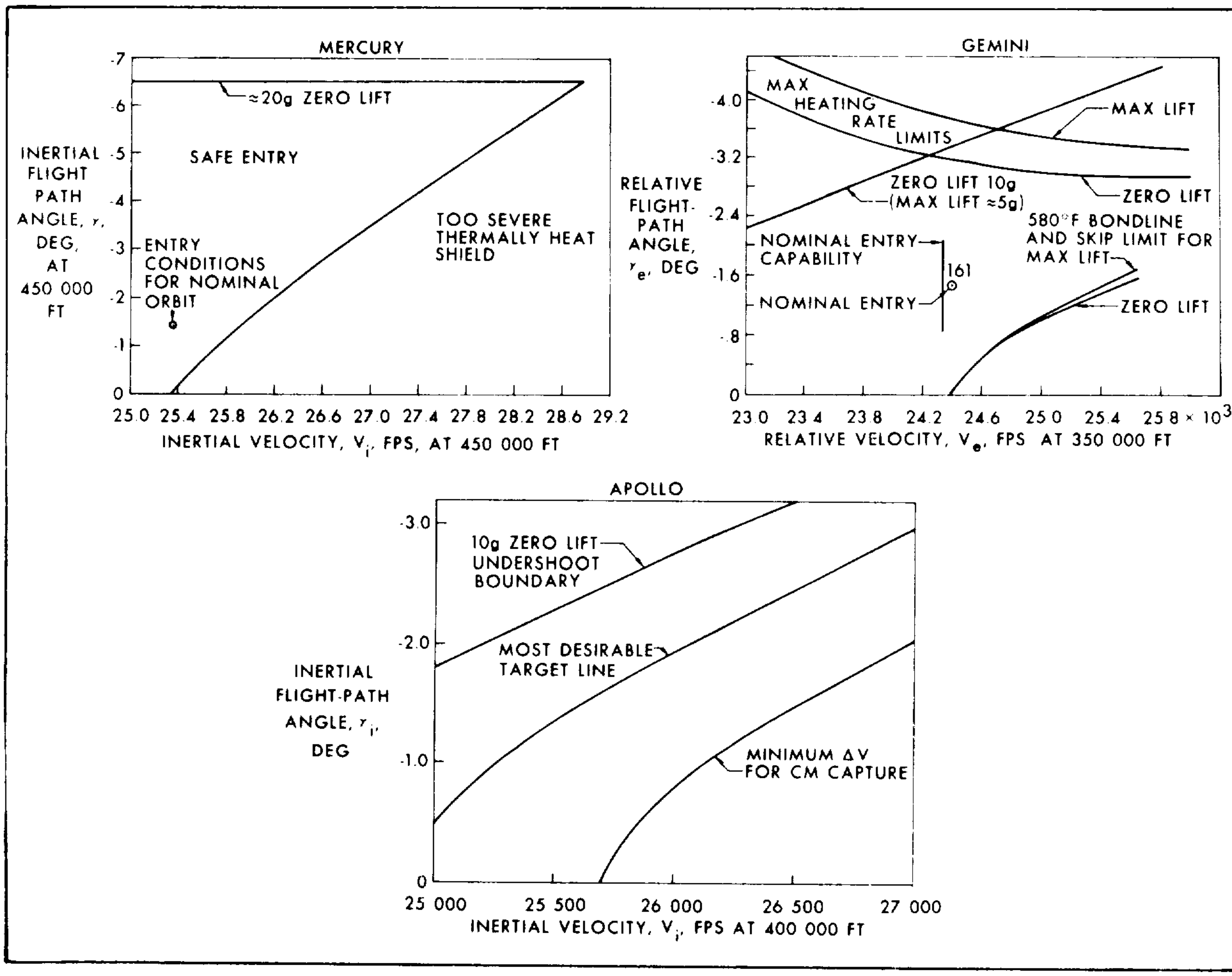

Figure 9.- Descent phase trajectory control limits.

limits were usually expressed in terms of velocity and flight-path angle at a particular altitude. As can be seen in figure 9, for Gemini relative conditions were used, whereas for Mercury and Apollo inertial conditions were and are being used. Although the limits are somewhat sensitive to the differences between relative and inertial conditions, the error in using one or the other is small considering the possibilities of entering at various azimuths and velocities and flight-path angles. In all three programs the limits basically have been determined by maximum desirable entry decelerations, thermal considerations, and atmospheric capture (skipout) considerations.

The deceleration limits were the highest for the Mercury spacecraft whereas the thermal limit was not much different for any of the spacecraft. The Gemini spacecraft had the additional heating rate constraints.

The Mercury entry conditions were basically restricted to the values shown in figure 9 since the retrorocket propulsive capability and orbital altitudes were essentially nonvariant. Thus, the deceleration limit did not act as a real constraint for that program.

The Gemini spacecraft entry conditions could vary considerably because of its orbital maneuver capability. The nominal entry conditions are shown for an elliptic orbit and a 161-n. mi. circular orbit. Circular orbit altitudes were restricted by the bondline or skipout limit, and elliptic orbit altitudes were restricted by all the limits shown.

The Apollo spacecraft propulsive and maneuver capability are such as to permit the planning of the deorbit maneuver to be along the desired target line. Thus, the entry limits are not constraining except for possible deorbit dispersions. Establishment of the limits is done in conjunction with the establishment of the desired entry target conditions. In all cases any dispersion or propulsive failures which cause the entry conditions to exceed any of the limits will call for use of one of the backup deorbit procedures.

It should be apparent that the contingency mission design involves almost all known constraints and is vitally necessary to insure crew safety as well as mission success. It should also again be stated that this part of mission design requires a large part of the total mission design effort and requires close coordination and communication between all elements involved in a particular program.

CONCLUDING REMARKS

The major constraints which must be considered during design of a mission to insure crew safety have been pointed out and are summarized in table VI. Flight or crew safety is one of the prime objectives in both nominal mission and contingency mission design. It has been pointed out that the contingency mission design and the "what-if" analysis involve considerably more effort than nominal mission planning. Examples have been shown of the limits and procedures which have resulted from this type of analysis and which are used in real time by the flight crew and ground flight control team to further enhance flight safety. In no case has it been necessary to sacrifice mission objectives for crew safety nor has it been necessary to sacrifice crew safety to achieve mission objectives. It has been possible to carry over proven philosophy from one program to another. It is believed that, through proper definition of mission-design-related constraints and through establishment procedures, sufficient flight safety for manned earth orbital missions has been achieved.

TABLE VI.- MAJOR MISSION DESIGN FACTORS

Mission Design
Flight Azimuth
Orbital Altitude
Descent Techniques
Contingency Mission Planning
Ascent and Descent Flight Corridor
Orbital Operations Planning
"What-If" Analysis the Major Effort
Simulation and Training Feedback

Although this paper has been limited to low altitude, earth orbital missions, the same philosophy and operational procedures are being applied to the various phases of the lunar landing mission. Analysis, to date, has shown this application to be valid.